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Meteorological Rocket Payload for Rohini-200, Its Developmental Details & Associated Physical Aspects

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An account of the work done at the Institute towards the development of a meteorological rocket payload to measure the temperature and wind in the altitude range 70-20 km is presented. Circuit details, data deduction and its accuracy using microbead thermistor as sensor are described. Various temperature corrections due to aerodynamic heating, solar radiation, longwave radiation, conduction along the support posts and thermistor time constant are theoretically calculated and discussed for the above mentioned altitude range. The use of thermistor mounts and their contribution towards the decrease of time constant of the sensor is discussed. The design and fabrication aspect of decelerators (parachutes) is also included.

1. Introduction

A rocket payload for measuring meteorological parameters, temperature and wind, is being developed at the Institute for the altitude range 70-20 km. It consists of an instrumented package which is ejected at the apogee from an indigeneous rocket launched at Thumba (TERLS) around 70 km, and falls suspended beneath a metallized silk parachute. During its descent it transmits, on a carrier link of 1680 MHz, information regarding the temperature of the atmosphere, together with the reference signal. The metallized parachute is tracked by a radar to obtain the trajectory and the wind, thus giving wind and temperature on real time basis.

2. Payload Instrumentation

Fig. 1 shows the elements of a temperature and wind measuring system. Fig. 2 shows the actual payload ready for testing in the rocket. The lowermost deck consists of battery pack and above it the electronic circuitry and telemetry system followed by the aerial and thermistor mount along with thermistor.

2.1 Battery Pack

Battery pack consists of three cells 6·3, 9 and 112V with current ratings 200, 2 and 35mA, respectively. These are dry type batteries regularly used for the radio sondes. For the purpose of payload they are moulded in an aluminium can with a potting compound, eccofoam. Results of the tests undertaken for various environmental conditions proved the suitability of their use in the rocket.

2.2 Electronic Circuit

The entire system utilizes 1680 MHz telemetry system using pulse amplitude modulation technique. Fig. 3 shows the block diagram of the circuit. The important part of the circuit is the resistance controlled oscillator (RCO)[1] where the frequency varies from 10 to 200 Hz corresponding to a thermistor resistance variation of 800 to 5kΩ for a temperature range of −80 to +25°C. A serious limitation on

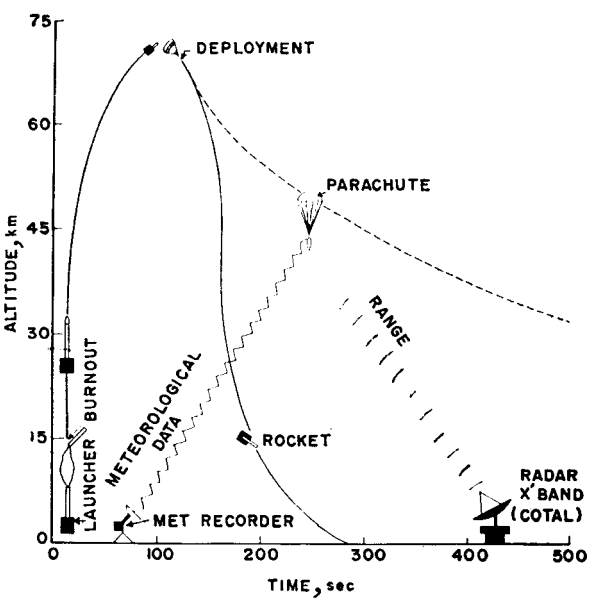

Fig. 1 — Complete wind and temperature measuring system

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the design of the oscillator is to limit the current passing through the thermistor so as to avoid the self-heating of the thermistor which is dictated by its dissipation constant. In the present type of thermistors used, dissipation constant varies from 45 to 90μW/°C.

During the course of development of payload, various circuits for RCO and commutator were tried. The final circuit being adopted in the payload is shown in the Fig. 4.

The NE555 is used as a timer (operated in multivibrator mode). The NE555 with proper RC combination makes 2N995 to conduct for 5 sec during which the reference resistance R_{ref} comes in the discharge path of the RCO, comprising unijunction 2N2646, and thus gives a signal of 190 Hz. During the non-conducting period of 55 sec of 2N995 reference resistance R_{ref} and thermistor (Th) are both in the discharge path of 2N2646 giving a signal (f_{th}) as determined by the total resistance of thermistor and reference resistance. The output of 2N2646 is shaped by another NE555 operated in univibrator mode to get a pulse width of 250 sec (This width is determined by the average power and duty cycle.) and fed to an amplifier and then finally to a cavity type 1680 MHz transmitter tube (RCA 6562). The tube RCA 6562 can deliver an average power of 400 mW at 1680 MHz.

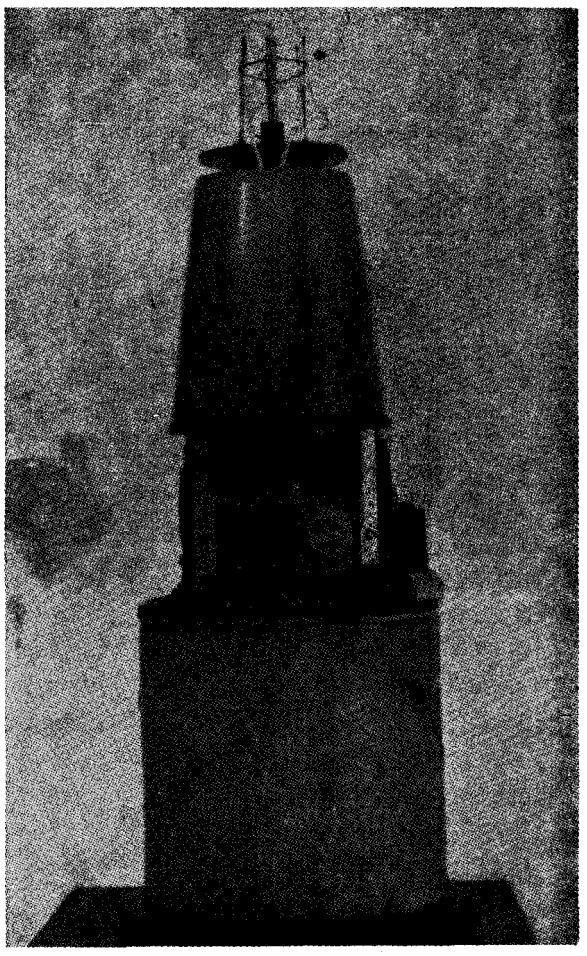

Fig. 2 — Payload ready for integration in the rocket

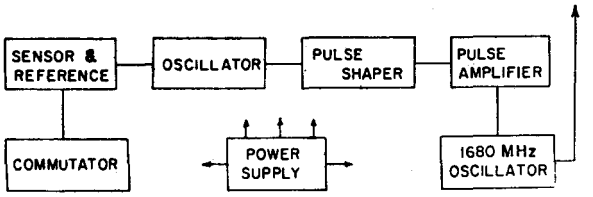

Fig. 3 — Block diagram of the payload electronic circuit

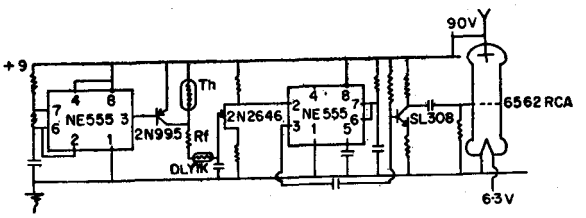

Fig. 4 — Present form of electronic circuit using NE555 timer (R_f stands for R_{ref}.)

Table 1 — Characteristics of the Two Types of Thermistors Used

Thermistor	Mass μg	Diameter cm	Time constant sec	Dissipation constant μW/°C
Veco	80	$2{\cdot}6 \times 10^{-2}$	0·5	545
Tempo	186	$3{\cdot}7 \times 10^{-2}$	0·8	90

The telemetry system is of pulse amplitude modulated type. On the ground, AN/GMD-1A receiver receives the telemetry, the signals being displayed on a 0-10 mV strip chart recorder having a paper width of 12". The parachute is tracked by the Cotal radar (3 cm) to get the winds.

2.3 Sensor

At present, two types of spherical microbead thermistors are being tried : (i) Veco (USA) 43A6 thermistor and (ii) indigeneous microbead thermistor (Tempo Ind., Bombay). The resistance variation for both thermistors is 5 kΩ to 800 kΩ corresponding to a temperature variation of +25 to − 80°C. Both the thermistors are spherical in geometry. Table 1 gives some of the characteristics of the thermistors. Fig. 5 shows the temperature versus resistance characteristics of both the types of thermistors. The thermistors were calibrated over a temperature range +30 to −60°C.

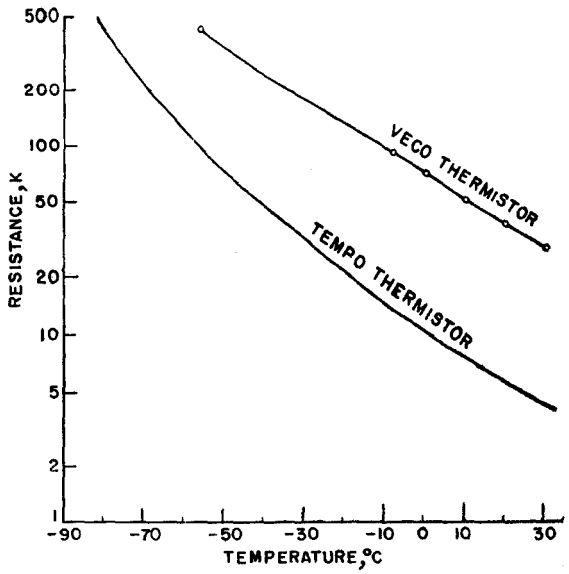

Fig. 5 — Characteristics of thermistors : (i) Veco and (ii) Tempo

3. Data Processing

The modulated frequency received on the ground varies from 10 to 200 Hz and a reference frequency of 195 Hz. As described earlier, for 55 and 5 sec temperature signals and reference signals are received alternatively.

Frequency generated by the above mentioned oscillators is inversely proportional to the discharge resistance (the condensor value of the oscillator circuit being kept constant). During the reference frequency (f_{ref}) only 68 kΩ (R_{ref}) is in discharge path. While in the case of signal frequency (f_{th}), the thermistor resistance and reference come in the discharge path ($R_{th} + R_{ref}$).

$$f_{ref} \propto \frac{1}{R_{ref}}$$

$$\therefore f_{ref} = \frac{K}{R_{ref}}, \quad K \text{ is circuit constant}$$

$$\text{Similarly, } f_{th} = \frac{K}{R_{th} + R_{ref}}$$

$$\therefore \frac{f_{ref}}{f_{th}} = \frac{R_{th} + R_{ref}}{R_{ref}}$$

$$\therefore R_{th} = \left(\frac{f_{ref}}{f_{th}} - 1 \right) . R_{ref}$$

Thus the determination of the R_{th} thermistor resistance entirely depends upon the ratio f_{ref}/f_{th} and R_{ref}, an advantage of introducing reference resistance.

Thus knowing R_{ref}, f_{ref} and f_{th} one can get the value of R_{th} (thermistor resistance) and going back to resistance versus temperature graph of the thermistor one gets the environmental temperature T_e. As regards the resolution, the modulating frequency (10-200 Hz) has a span of 200 Hz corresponding to a temperature variation of —80°C to +30°C, roughly a span of 100°C. Thus 2 cycles correspond to 1°C variation. On the recorder 0-10mV (Westronix), there are 100 divisions for recording 200 c/s. There one can easily read ½ division corresponding to 1 cycle. Thus the end resolution is 0·5°C. The resolution is limited by the modulating frequency and recorder width. Fig. 6 shows the record obtained during a ballon flight.

Fine variations of temperature signals can be seen in the record which are usually not recorded in the routine radiosonde. This is due to the small size of the thermistor.

4. Radiative Heating Processes for the Thermistor

The behaviour of temperature sensing element can be estimated by knowing the energy exchange processes between the sensing element and the environment. The equation of heat flow for the thermistor is[2]

$$\frac{dQ}{dt} = C_d + C_v + R_t + W$$

$$= m_t C_t \frac{dT_t}{dt}$$

where

$$Q = H_t = \text{Total input energy}$$

$$C_d = \frac{2K\beta}{l}(T_s - T_t)$$

$$C_v = h_t A_t (T_e - T_t) + h_t A_t . rV^2/2C_p$$

$$R_t = \frac{JA_t\epsilon}{4} + A_t \sigma a T_e^4$$

and

W Electrical heating power (minimized by the circuit design) neglected in the present case

m_tC_t Heat capacity of the thermistor

m_t Mass of the thermistor

C_t Specific heat of the thermistor

C_d Heat input due to the conduction through the lead wires

K Thermal conductivity of lead wires

β Cross-sectional area of lead wires

l Length of lead wires

T_s Temperature of the thermistor posts

T_t Temperature of the thermistor

C_v Heat input due to forced convection = $h_t A_t (T_e - T_t) + rV^2/2C_p . h_t A_t$

In the above form of C_v, the first term is due to forced convection and the second term is due to the aerodynamic heating .

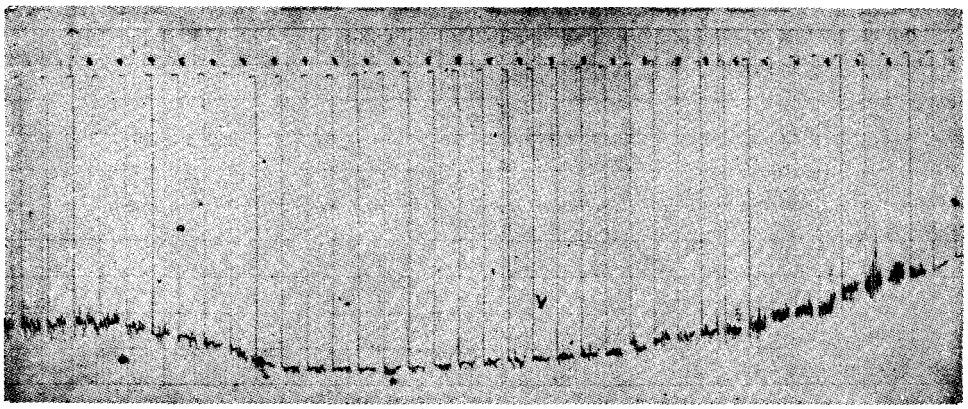

Fig. 6 — A ballon flight record taken with payload (Fine variations of temperature are recorded along with regular reference signals)

And

h_t Convective heat transfer coefficient

A_t Thermistor area

r Recovery factor

V Speed of the thermistor, the fall velocity of the parachute

C_p Specific heat at constant pressure

R_t Heat input due to radiation $= JA_t\epsilon/4 + A_t\sigma a T_e^4$

In the above expression for R_t, the first term represents the contribution due to solar radiation and second term represents long wave radiation (taken as negative). In fact this term has to include the outgoing radiation due to the payload itself. This is neglected in the present computation. We have

J Solar constant

ϵ Absorption coefficient for solar radiation

σ Stefan-Boltzmann constant

a Absorption coefficient of thermistor

$A_t/4$ Projected area of the thermistor

Putting all terms together we have

$$m_t C_t \frac{dT_t}{dt} = h_t A_t (T_e - T_t) + \frac{rV^2}{2C_p} + \frac{2K\beta}{l}(T_s - T_t) + \frac{JA_t\epsilon}{4} - A_t \sigma a T_e^4 \quad ...(1)$$

But $T_s = T_e + \Delta T_{se}$ where ΔT_{se} is the temperature difference between support post and the environment

$$\therefore m_t C_t \frac{dT_t}{dt} = h_t A_t \left\{T_e - T_t + \frac{rV^2}{2C_p}\right\} + \frac{2K\beta}{l}\left\{T_e - T_t + \Delta T_{se}\right\} + \frac{JA_t\epsilon}{4} - A_t \sigma a T_t^4 \quad ...(2)$$

In Eq. (2) the left-hand side gives the change in the heating to thermistor and the right-hand side gives the causes therefor. Eq. (3) is nonlinear because of the presence of T_t^4, but can be simplified. Supposing

$$\theta = T_t - T_e$$

Then

$$T_t = T_e + \theta$$

$$\therefore T_t^4 = T_e^4 + 4T_e^3\theta + \text{higher order terms}$$

Neglecting higher order terms, we have

$$m_t C_t (dT_t/dt) = \left\{h_t A_t \frac{rV^2}{2C_p}\right\} - \theta\left\{h_t A_t + \frac{2K\beta}{l} + 4\sigma A_t T_e^3\right\} + \frac{2K\beta}{l}\Delta T_{se} + \frac{JA_t\epsilon}{4} - A_t \sigma a T_e^4 \quad ...(3)$$

$$\therefore \theta = \frac{-m_t C_t \frac{dT_t}{dt} + h_t A_t \frac{rV^2}{2C_p} + \frac{2K\beta}{l}\Delta T_{se} + \frac{JA_t\epsilon}{4} - A_t\sigma a T_e^4}{h_t A_t + 4\sigma A_t T_e^3 + \frac{2K\beta}{l}} \quad ...(4)$$

In the numerator, the first term gives the correction due to the time constant of the thermistor; dT_t/dt is negative while the payload comes down and hence the complete term becomes positive. And, $\frac{dT_t}{dt}$ is found to be maximum of the order of 2·2°C/sec (Ref. 2). The second term is mainly due to the aerodynamic heating of the thermistor. This term becomes important over 40 km, above which the speed of the parachute is of the order of speed of sound. The third term is due to the conduction between mounting posts and the thermistor. At the time of ejection of the payload the posts have already attained the nosecone temperature which may be 100°C more than the environmental temperature. Thus there is a

temperature difference of 100°C between thermistor and thermistor posts at the apogee. This difference decreases as the payload comes down. Thus this term contributes more between 70 and 40 km. Variation of ΔT_{se} are adopted from Arcus rocket data for present computation. The fourth term is due to the solar radiation falling over the thermistor. As there is no shield over the thermistor, it may get heated by the direct solar radiation. This term can be neglected if the flight is undertaken in the night. The term $A_t \sigma a T_e^4$ due to long wave radiation of surrounding is neglected, being very small in magnitude compared to other terms. The denominator is called the dissipation factor of the thermistor. The time constant of thermistor is defined by

$$\tau = \frac{m_t C_t}{h_t A_t + 4\sigma A_t T_e^3 + 2K\beta/l}$$

$$= \frac{\text{Heat capacity of the thermistor}}{\text{Dissipation factor of the thermistor}}$$

Fig. 7 gives the variation of constant of Veco and Tempo thermistors.

5. Usage of Mounts

As shown in Fig. 7 the time constant of the thermistor is considerable between 30 and 80 km. The time constant at 70 km is of the order of seconds. During this time the parachute descends by a few kilometres leading to an error in the determination of temperature. Hence it is desirable to reduce the the time constant. The STS mount[2] (currently used in USA) has a mylar film mounted on support posts and the thermistor is fixed over it. The mass of the film is comparable to that of thermistor. But the exposed area is much more than the thermistor area which allows the above mentioned dissipation

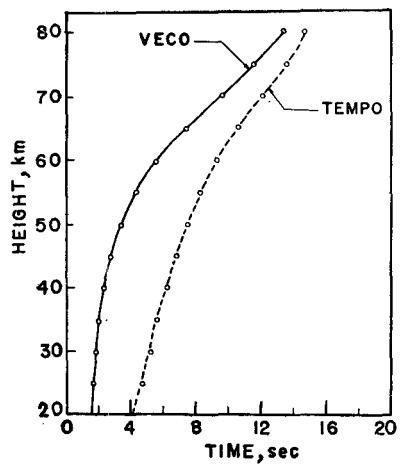

Fig. 7 — Variation of time constant of thermistors (Veco and tempo) with altitude, computed using equation for τ

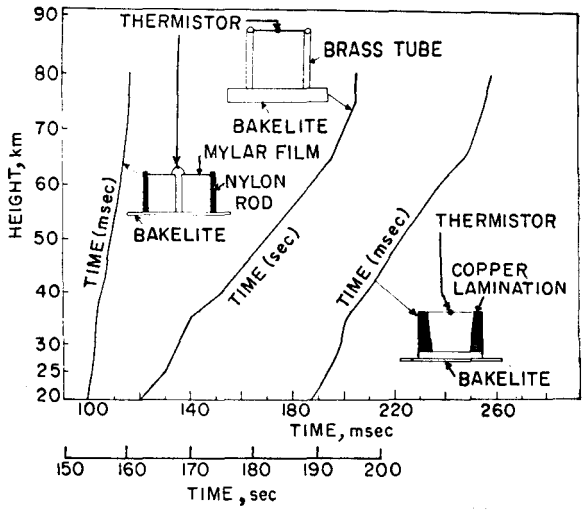

Fig. 8 — Variation of time constant of various mounts

factor to increase leading to a faster heat transport process to the thermistor thus reducing the value of time constant. IITM has tried a few mounts. One of them has a bakelite base with the unwanted copper portion etched out. The response of this mount is shown in Fig. 8 along with STS mount. In Fig. 8 the response of another laboratory type mount is also shown whose time constant is of the order of sec. Hence it is not recommended for flight purposes. The final corrections incorporating time constants of these mounts are not included in the present report.

6. Computation of Errors

To find out the difference in $T_t - T_e$ in Eq. (4) all constants are known except h_t. The value of this quantity, viz. $T_t - T_e$ computed in the following way.

Some of the dimensionless parameters regularly used in the fluid dynamics for the various investigations are as follows.

Raynolds number $= Re = \rho\, Vd/\mu$

Prandtl number $= Pr = C_p \mu/K$

Mach number $= M = V/a$

Nusselt number $= Nu = h_t d/K$

where

ρ Density

V Fall rate of Skua type parachute

d Diameter of the thermistor

μ Coefficient of viscosity

C_p Specific heat at constant pressure

K Thermal conductivity

a Velocity of sound

h_t Heat transfer coefficient (convective)

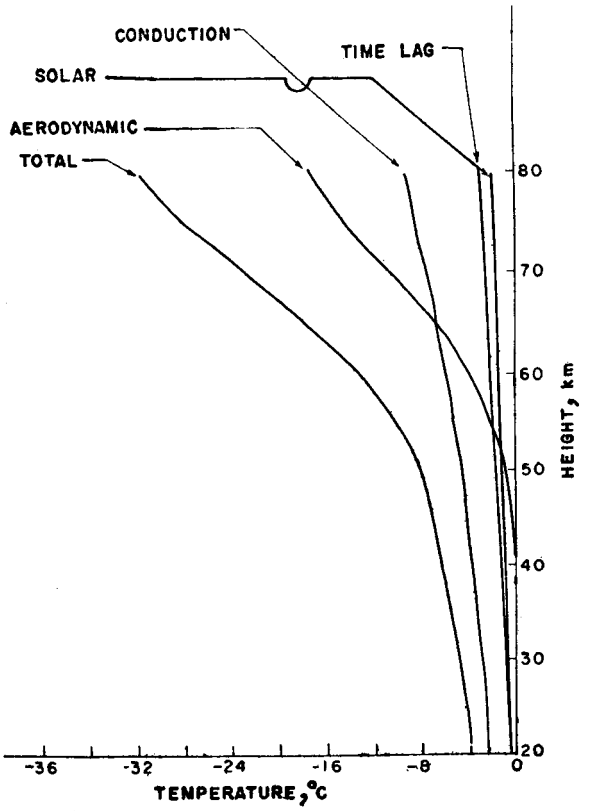

Fig. 9 — Variation of computed errors with altitude

In problems involving heat transfer, the main quantity of interest is the heat transfer coefficient h_t expressed in terms of Nusselt number Nu

$$h_t = \frac{Nuk}{d}$$

If we consider the shape of the thermistor to be spherical, then the Nu will be (in continuum region)[3]

$$Nu_c = 2 + 0{\cdot}37\ (Re)^{0{\cdot}6}.\ (Pr)^{1/3}$$

and for slip region :

$$Nu_s = \frac{Nu_c}{1 + \frac{3{\cdot}42\ M.\ Nu_c}{Re.\ Pr}}$$

Thus once Nu is known, h_t can be calculated for various altitude levels. With all the known factors including h_t, the dissipation factor ($h_t\ A_t + 4\sigma\ A_t\ T_c^3 + 2K\beta/l$) for the thermistor and for various types of mounts can be calculated.

With the dissipation factor and the thermal capacity of each body, their respective time constants can also be calculated.

Fig. 9 shows the variation of different errors to be applied to the raw temperature data. Earlier these corrections were calculated[4] manually for a few heights. Now the complete correction procedure is computerized. The corrections can be obtained at any level at an interval of 1 km, feeding the characteristics of the thermistor and the fall rate of parachute to the computer. A 1968 US standard atmosphere data for density (ρ), viscosity (μ) and thermal conductivity K are used for the calculations.

7. Parachute

This is an important part of the payload determining the fall rate of thepayload. At the apogee of the rocket the payload along with attached parachute is ejected and for the first few seconds (corresponds to a height of few kilometres) the whole system has a free fall. Then the parachute opens and the payload descends slowly thereafter. Design consideration of the parachute require a low ballistic coefficient equal to W/C_DA, C_D drag coefficient, W the weight of parachute and A the cross-sectional area of the payload. The drag coefficient is maximum when the ratio of inflated area of parachute to Fabric area (A_i/A_f) is of the order of 0·5.

The fall velocity of parachute is given by $U = \sqrt{\frac{2mg}{\rho C_D\ A_f}}$ where ρ is air density, m the mass of parachute, A_f the fabric area. Other than its physical characteristics the fall rate of the parachute is inversely proportional to the square root of air density. Fig. 10 shows the fall rate of the IITM parachute along with Skua type parachute.

For the present project a parachute with alternate segments of mylar film and silk are utilized with a total fabric area of 6m². In the later models the silk

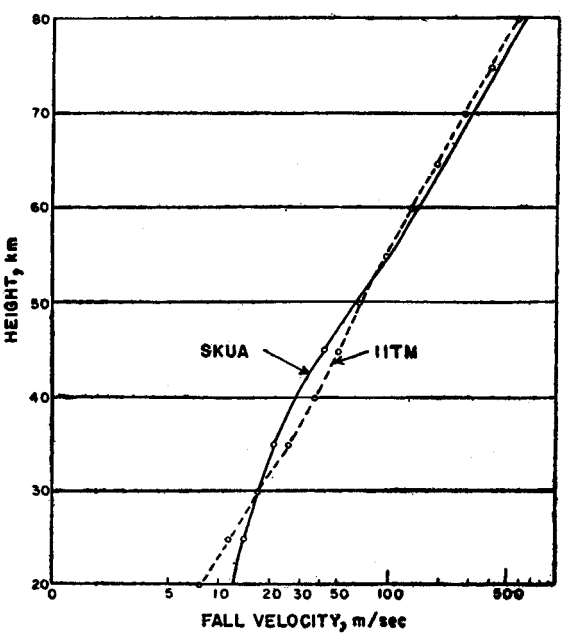

Fig. 10 — Fall rate of Skua and IITM parachute (The fall rate of Skua type parachute is used for computing the various errors in the text)

was replaced by mylar film thus giving more reflecting area ($12m^2$). In fact for a proper radar reflection an area of $60m^2$ is required. But due to the limited space available in the rocket for packing the parachute, a compromise is made towards above mentioned reflected area.

8. Facilities Available at Thumba

At present the Thumba Equatorial Rocket Launching Station (TERLS) is in a position to supply Rohini-200 rockets for the meteorological payloads. Rohini-200 is a two-stage rocket with a capacity to carry a payload of 2·2 kg up to a height of 70 km when launched at 75-80° elevation.

TERLS has a ground support division to carry out the qualifying tests on the payload. The qualifying tests, include (i) shock test, (ii) vibration test, (iii) spin test, (iv) temperature test and (v) vacuum test. The payload has to undergo all these tests in hot (battery on) conditions. Some important outputs of the payload are monitored before and after each test.

The GMD receiver is available for receiving the temperature signals. For tracking the parachute, a Cotal (3cm) radar is used.

9. Conclusion

The IITM payload is complete in all respects; including the design and the environmental tests. Earlier eight flights on Menaka II did not come through for various technical reasons. Now it is planned to finalize the payload using Rohini-200 rocket.

Acknowledgement

Thanks are due to Sarvashree L K Sadani, Prem Prakash and R N Sengupta who were associated during the initial stages of the development of payload. Ground support of TERLS especially that of Project Manager, RSR, Mr V Sudhakar is also acknowledged. Thanks are also due to Shri H Mitra who gave initial impetus to this project.

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